

NASA Technical Memorandum 104180

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# THE BENCHMARK AEROELASTIC MODELS PROGRAM - DESCRIPTION AND HIGHLIGHTS OF INITIAL RESULTS

(NASA-TM-104180) THE BENCHMARK AEROELASTIC  
MODELS PROGRAM: DESCRIPTION AND HIGHLIGHTS  
OF INITIAL RESULTS (NASA) 13 p CSCL 01A

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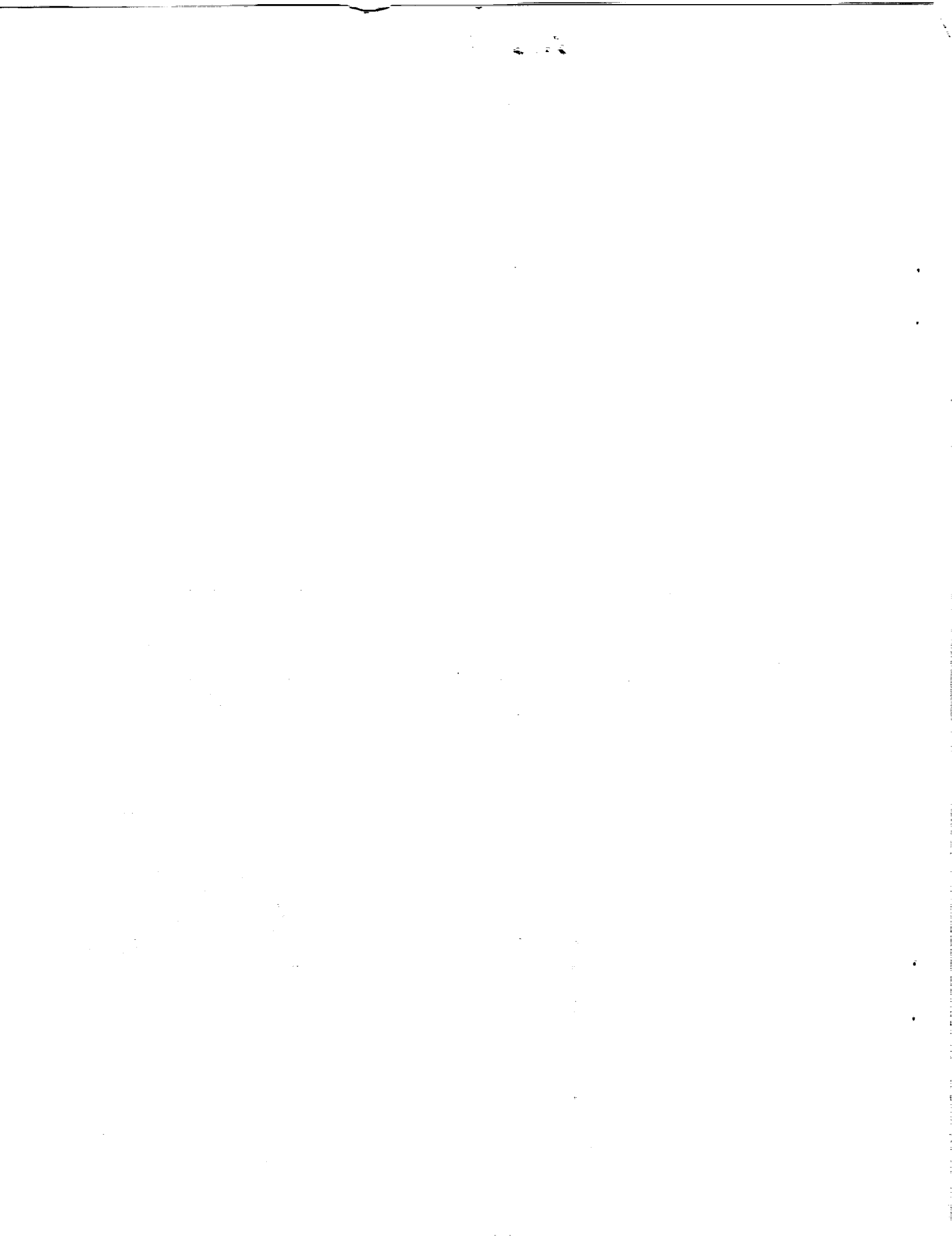
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December 1991

**NASA**

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# THE BENCHMARK AEROELASTIC MODELS PROGRAM — DESCRIPTION AND HIGHLIGHTS OF INITIAL RESULTS

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## 1. SUMMARY

The Structural Dynamics Division of NASA Langley Research Center has implemented an experimental effort in aeroelasticity called the Benchmark Models Program. The primary purpose of this program is to provide the necessary data to evaluate CFD codes for aeroelastic analysis. It also focuses on increasing the understanding of the physics of unsteady flows and providing data for empirical design. This paper gives an overview of this program and highlights some results obtained in the initial tests. The tests that have been completed include measurement of unsteady pressures during flutter of a rigid wing with a NACA 0012 airfoil section, and dynamic response measurements of a flexible rectangular wing with a thick circular arc airfoil undergoing shock-boundary layer oscillations.

## 2. INTRODUCTION

A significant number of aircraft aeroelastic problems occur in the transonic speed range. Generally, minimum flutter speed is encountered at transonic Mach numbers. In addition, buffeting, control surface buzz, and other non-classical instabilities may be encountered. Computational fluid dynamic (CFD) computer codes are now maturing and hold promise for rational analysis of all these phenomena. The state of the art in this area is reviewed by Edwards and Malone.<sup>1</sup>

Currently, the assessment of the CFD codes even for the classical flutter problem is far from complete. For example it is not clear which equation level is required for a given configuration, Mach number, and angle of attack range. One reason for this situation is the level of resources required to apply the CFD codes for enough cases to establish trends. Typically these codes require enormous computer resources even to evaluate one flutter boundary, and also require significant expertise and effort by the users. However, an additional and very significant reason for the incomplete calibration of the CFD codes is the lack of well documented experimental data sets.

Although the flutter data available in the literature is quite extensive, much of it is not suitable for validation efforts. For example, after an extensive literature search, only one configuration was accepted as an AGARD standard configuration<sup>2</sup> and the calculation of mode shapes from a finite element model was required. Early experimenters were operating within a framework of linear theory which does not require airfoil shape, for example, and airfoil ordinates were not generally measured.

Similarly, modal definitions or model structural and mass properties were given within a framework of beam theory. In addition, many of the investigations give only the flutter boundary defined in terms of the test conditions such as dynamic pressure and Mach number at flutter, sometimes even omitting the flutter frequency. Such data sets are useful as a guide for CFD validation, but they provide little insight in the event of discrepancies which are at times encountered. Reliance must then be placed on the experience and intuition of the investigator to resolve the problems encountered in applications. Such is particularly the case with CFD codes as it is very difficult to separate numerical shortcomings and the limitations of the treatment of the flow physics. It is very difficult to evaluate convergence in terms of the computational grid or time step within computer budgets, time, and memory constraints and questions concerning the numerical solutions are seldom answered. For example, premature rises in the flutter boundary versus Mach number are sometimes encountered. The premature rises may be related to an inadequate computational grid, but also may be related to the required equation level or other factors.

There are many significant data sets available for measured unsteady pressures on models undergoing forced oscillations. Such data are, of course, fundamental to the validation of CFD codes, but it is difficult to assess the implication of discrepancies between calculated and measured experimental pressures for flutter analyses.

The Structural Dynamics Division (SDyD) of the NASA Langley Research Center has been actively involved in the development and application of CFD codes for treating the flutter problem for nearly two decades. In view of the difficulty that has been experienced in evaluating such codes in comparison with current data sets, an experimental program in aeroelasticity has been developed and is called the Benchmark Models Program. The primary purpose of this program is to provide well documented data sets suitable for CFD code validation. Additional supplementary goals are to provide increased understanding of the physics of transonic unsteady flows, and where necessary provide data for empirical design. This paper gives an overview of the SDyD Benchmark Models Program, describes the models for the tests, and then gives highlights of some of the initial tests.

## 3. BENCHMARK MODELS PROGRAM OVERVIEW

The SDyD Benchmark Models Program is a joint effort of



Figure 1. PAPA model with NACA 0012 Airfoil mounted in TDT.

the three aeroelasticity-related branches of SDyD, the Configuration Aeroelasticity Branch, the Unsteady Aerodynamics Branch, and the Aeroservoelasticity Branch. It consists both of simple models for concept exploration, and highly instrumented models for CFD validation studies. The test team consists of about six engineers, depending on the test, with varied backgrounds such as wind tunnel testing, CFD applications, and control systems. The testing is being conducted in the NASA Langley Transonic Dynamics Tunnel (TDT), and is scheduled for about two tests per year.

Goals for the benchmark models for CFD validation studies include:

- Aerodynamically smooth surfaces
- Complete description of geometry including static and dynamic deformation
- Complete experimental definition of structural dynamics including modal frequencies, dampings, generalized masses, and mode shapes
- Measured flutter boundary including flutter frequency and mode shapes
- Measured unsteady pressures on at least two chords during flutter

- At least qualitative indication of transition and separation
- Flow visualization where possible

The testing program has been designed to start with simple models and then to evolve into more complex models and tests. This is advantageous from the test technique development point of view as well as for CFD validation. The initial tests are for rigid wings mounted on the pitch and plunge apparatus (PAPA). The wings are rectangular in planform and are of panel aspect ratio 2.0. The initial wing is shown in figure 1 as mounted in the Transonic Dynamics tunnel. Currently there are three wings with different airfoils in this series for conventional flutter testing. These three models are designed to be essentially plug-compatible for ease of testing, instrumentation, and data processing. In addition, a model similar to one of this series will be tested with a trailing edge control and upper and lower surface spoilers. Active flutter control systems will also be tested using this model. A flexible high speed civil transport (HSCT) model is also incorporated into the plan following several of the initial tests. Subsequent models will investigate other widely-varying planforms.

The test plan is illustrated in the tentative schedule shown in figure 2. At this time two models have been tested. One was a simple model to briefly investigate the dynamic response of a flexible wing with an 18% circular-arc airfoil undergoing periodic shock-boundary layer oscillations.<sup>3</sup> This model was tested in the spring and fall of 1990 as shown in figure 2. The first of the models on the PAPA mount system had a NACA 0012 airfoil<sup>4</sup> (fig. 1) and was tested in the summer of 1990 and winter of 1991. Some highlights of these tests will be subsequently presented after further description of the PAPA system, the wind tunnel, and the Benchmark Models.

#### 4. THE PAPA MOUNT SYSTEM

As previously indicated, several of the Benchmark Models are to be tested on the Pitch and Plunge Apparatus (PAPA) of the Langley Transonic Dynamics Tunnel (TDT).<sup>5,6</sup> A photograph of the PAPA mount is shown in figure 3. It consists primarily of four steel rods attached to a turntable on the wall of the tunnel and attached to a moving steel plate at the other end (fig. 3). The rods permit vertical translation or plunge, and pitch or torsional motion. A central beam that is thin vertically, but wide horizontally, stiffens the system in the fore-and-aft direction. The turntable is remotely adjustable to permit changes in angle of attack. The rods have essentially fixed-fixed

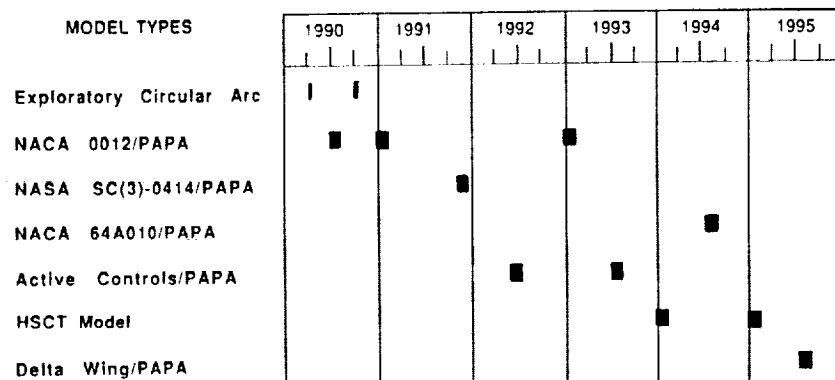


Figure 2. Benchmark Models test schedule.

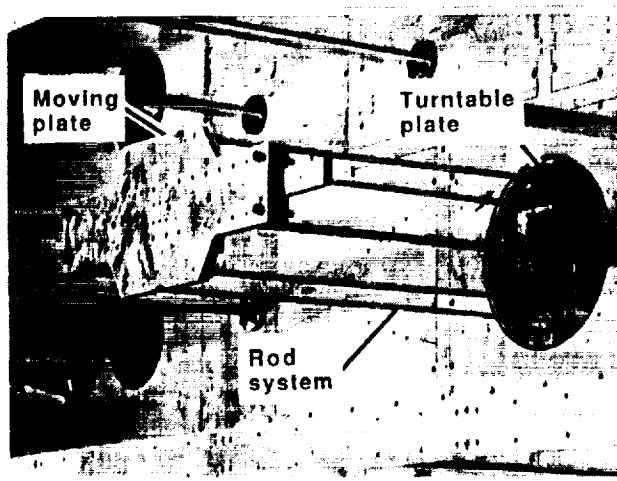


Figure 3. Photograph of pitch and plunge apparatus (PAPA) mounted in the TDT.

end conditions to provide linear pitch and plunge stiffnesses for elastic restraint. The PAPA mechanism is instrumented with strain gages to provide pitch and plunge position, and accelerometers to measure pitch and plunge accelerations. The root of the wing is attached to the moving plate in the wind tunnel.

As shown in figure 3, the PAPA system projects out into the wind tunnel. A splitter plate is used as an effective end plate or side wall. This splitter plate is 10 feet high (3.05 m) by 12 feet long (3.66 m) and is shown in figure 4. The center of the PAPA system is 7 feet (2.13 m) from the leading edge of the splitter plate. An end plate attached to the root of the model covers the hole through which the wing mounting pedestal extends. This circular end plate is one chord in diameter and is recessed into the splitter plate. The splitter plate is supported from the wall by struts that extend about 3.3 feet (1.0 m) from the wind tunnel wall. The PAPA mount system is surrounded by a steamlined fairing behind the splitter plate. For the Benchmark Model tests, splitter plate pressures are measured with 20 pressure transducers (fig. 4). A 0.42 foot (0.13 m) span boundary layer rake with ten pressure transducers is located above and aft of the wing. Studies are currently underway to examine the feasibility of locating the PAPA system behind the tunnel wall to simplify installation.

The PAPA system is quite rugged and robust thus permitting measurement of many flutter points with very low risk to the models. The strength of the system permits flutter testing at moderate angles of attack unlike the usual flutter models which are limited to small values by aerodynamic loads. Most models tested on PAPA have a somewhat mild flutter crossing which permits dwelling at nearly constant amplitude for even as long as one to two minutes so that many cycles of data can be used for averaging measured pressures. The natural frequencies of this system are usually around three to five Hertz which also permits easier flutter testing than for models with higher frequencies. The PAPA system contains no bearings, and the structural damping is very low, on the order of 0.0005 in fraction of critical damping. Overall the mount system can be well defined such that the effect of unsteady aerodynamics on flutter can be investigated in detail.

For static pressure measurements, the system can be rigidized

readily with a simple fixture. For later tests, installation of a strain gage balance for steady state force measurements is being investigated. The development of an excitation system to permit dynamic measurements prior to flutter is also underway.

The moving plate and rods of the PAPA are relatively heavy. The models can therefore also be relatively heavy without significant further penalty. Thus it is practical to use machined metal models along the lines of an aerodynamic static test model. These models can be very smooth by usual aeroelastic model standards, and can be manufactured much less expensively than the usual flutter model. A smooth surface is considered vital for transonic benchmark aerodynamic data.

One consequence of the large mass of the PAPA/model system is that the flutter data is for high mass ratio, on the order of 1000 in air (or 250 in the heavy gas). This leads to an unusually low value of reduced frequency,  $k$ , of the order of 0.02 based on semichord (in air). Such a low reduced frequency would normally be expected to accentuate transonic aerodynamic effects.

## 5. WIND TUNNEL

The Benchmark Model tests are to be conducted in the Langley Transonic Dynamics Tunnel (TDT). This tunnel is a large facility with a test section 16-foot (4.88 m) square with cropped corners. All four walls are slotted. The TDT is a continuous flow, single return tunnel that can operate at Mach numbers up to 1.2, and for pressures from near vacuum to atmospheric. Either air or a heavy gas can be used as a test medium, but only air has been used for the initial Benchmark Model tests. This tunnel is used primarily for aeroelastic testing, and is equipped with four quick-opening bypass valves for rapidly reducing test section dynamic pressure and Mach number upon encountering an instability. The large tunnel size and the use of heavy gas as the test medium considerably facilitate aeroelastic model design and instrumentation.

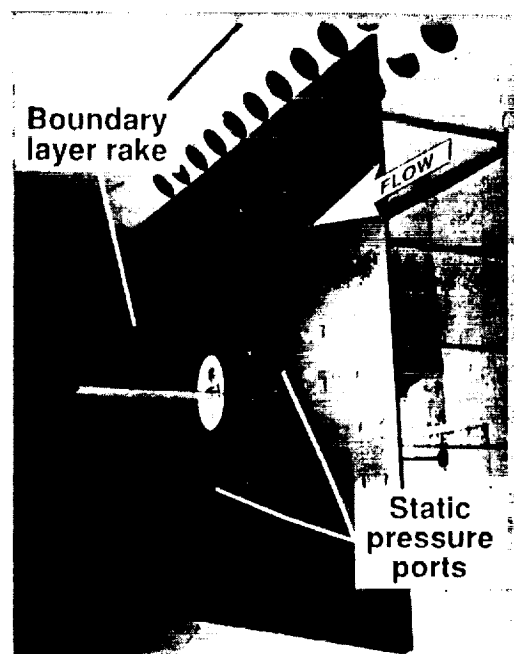


Figure 4. Splitter plate arrangement for PAPA tests.

A key ingredient in the Benchmark Model tests is the data acquisition system of the TDT. The PAPA models are currently designed for 128 channels of data, with later models increasing to 192 and 256 channels. Software has been developed to permit nearly on-line display of first harmonic and static data. Typically 40 seconds of data are recorded at 100 samples/second for nearly on-line analysis. For subsequent analyses, 20 seconds of the time history of each data channel is recorded at 1000 samples/second. These data become a massive set of data for a typical test and are recorded on tape. Transfer to central site supercomputers has been accomplished. Data gathering, handling, reduction, and analysis for tests of this type is a large effort and requires considerable specialized software development. This data processing system is still being developed and refined for the Benchmark Models program.

## 6. DESCRIPTION OF PAPA MODELS

### 6.1 Conventional Flutter Models

The first Benchmark Model for the PAPA system is shown in figure 1. As previously mentioned, there are three similar models in this series that differ only in airfoil section and are designed for basic flutter tests. The three airfoils are the NACA 0012, the NASA SC(2) 0414, and the NACA 64A010. The profiles of these airfoils are shown in figure 5. The NACA 0012 is an old design, twelve per cent thick airfoil that has been extensively tested. For example, reference 7 summarizes over forty steady wind tunnel tests for this airfoil. The NASA SC(2)-0414 is a typical modern supercritical airfoil and is described as one of a series of airfoils in reference 8. It has a design lift coefficient of 0.4, is fourteen percent thick, and is described as an airfoil for a business jet<sup>8</sup>. The NACA 64A010 is a symmetrical ten percent thick NACA design that has been used in an AGARD standard unsteady two dimensional pressure test,<sup>9</sup> and in a three dimensional test<sup>10</sup>. These three airfoils have very different types of transonic flow development. The NACA 0012 airfoil develops a shock wave forward of midchord as Mach number is increased into the transonic range. The SC(2)-0414 is an aft-loaded supercritical airfoil with significant aft

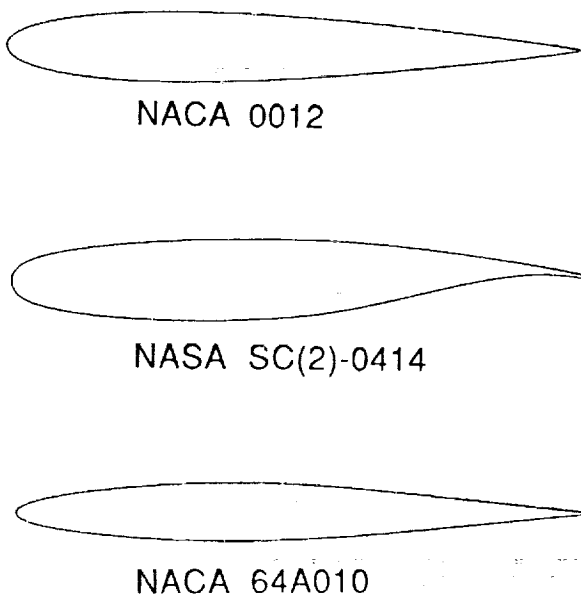


Figure 5. Airfoils for initial PAPA wings.

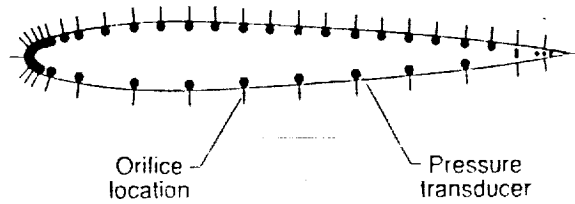


Figure 6. Orifice and pressure transducer locations for 0012/PAPA model.

camber and develops a shock further aft. The NACA 64A010 airfoil is somewhat intermediate. Although the PAPA models are of relatively low aspect ratio, this range of airfoils should give a good survey of the effects of widely differing airfoils on transonic flutter characteristics for CFD calibration studies.

As shown in figure 1, the PAPA models are rectangular in planform. They have a 16 inch (0.406 m) chord and a semispan of 32 inches (0.812 m) plus the tip of revolution. There are two rows of in situ pressure transducers, each row containing 40 unsteady pressure transducers. One row is at 60 per cent span, and the other one at 95 per cent span. The location of the pressure transducers for the 0012 model is illustrated in figure 6. The model is machined from aluminum and is constructed in three sections that are bolted together. A row of orifices is located about one inch (2.54 cm) outboard of each of the outer joints. The pressure transducers were bonded into brass tubes for protection during installation and removal, and the brass tubes were bonded into holes drilled into the wing section. The mounting holes, a bare transducer, and a transducer mounted in a brass tube are shown in the upper left portion of figure 7. Four accelerometers near the corners of the wing were installed in pockets as shown in the upper right portion of the figure.

During the initial test of the NACA 0012 model in July 1990, only the inboard row of transducers was installed, but both rows were operational during the January 1991 tunnel entry.

The model with the NASA SC(2)-0414 airfoil has been completed and is being prepared for testing during November and December 1991 (fig. 2). The model is constructed in essentially the same fashion as the 0012 model with only some minor improvements in detail. It is designed to be essentially plug compatible with the 0012 model. Some redistribution of

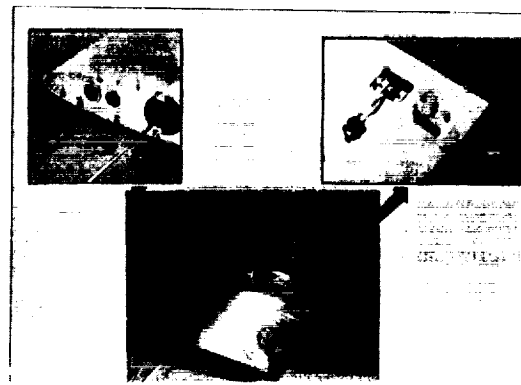


Figure 7. Details of 0012 model.

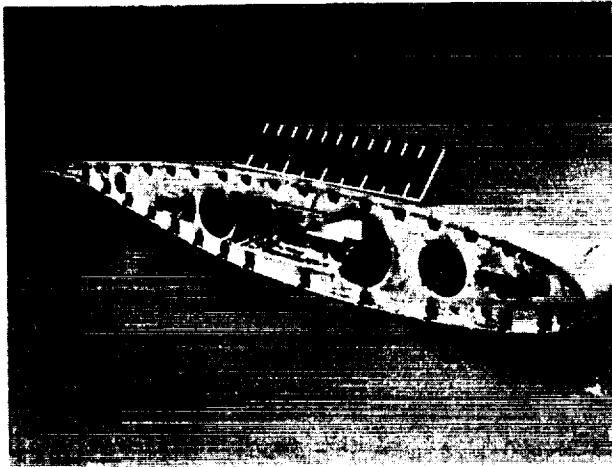


Figure 8. Cross section of SC(2)-0414 model showing transducer locations.

the pressure transducers has been made by moving some from the nose to the aft lower surface to improve the definition of pressures in the aft lower surface. A photograph of a section of this model is shown in figure 8. The holes near the surface are for mounting the pressure transducers.

The model with the NACA 64A010 airfoil has been designed and is being machined. It is scheduled for a later entry (fig. 2), but may serve as a backup for the other tests if mechanical or instrumentation problems are encountered with the other models.

### 6.2 Active Controls Model

An active controls model is under construction to investigate flutter suppression on the PAPA system.<sup>11</sup> This model will have a NACA 0012 airfoil and will be very similar to the other NACA 0012 flutter model in order to build on the experience and results of the earlier model. The planform and controls layout are shown in figure 9. The model will have a thirty per cent span trailing edge control of twenty five percent chord. Spoilers are located on the upper and lower surfaces of the wing upstream of the trailing edge control. The spoilers are fifteen per cent chord in length. The unsteady pressures will be measured at one full chord which is the same as for the earlier model but with a different distribution to define the pressures near the hinge lines of the control surface and spoilers. An additional partial row of pressure transducers is located at

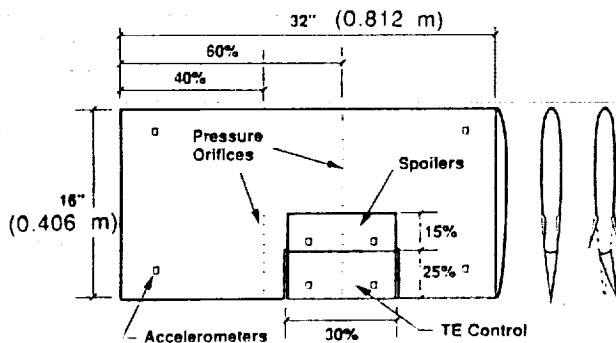


Figure 9. Drawing of active controls model.

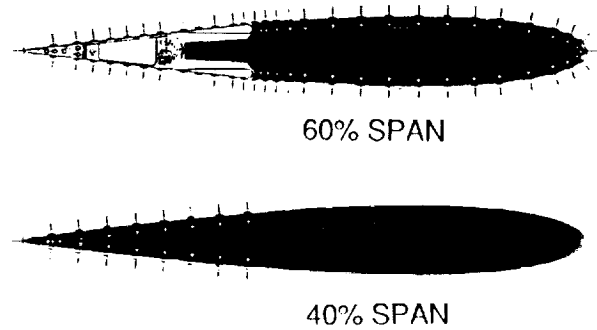


Figure 10. Orifice locations for active controls model.

forty per cent span (fig. 9). The planned orifice locations are presented in figure 10.

To meet the space and torque requirements for this model, a new hydraulic actuator is being designed. A prototype actuator has been built and is being tested. The breadboard test setup is shown in figure 11. Laboratory tests to determine the dynamic characteristics and load limits are underway.

Two tunnel entries are planned (fig. 2) for this model. The initial entry will measure the open-loop flutter boundaries for comparison with results from the earlier model. The model will also be mounted on a five component force balance which will permit measurement of the static and dynamic loads of the model with oscillating controls. The experimental data base will be used to design active flutter suppression control laws. The second entry will evaluate these control laws.

### 7. HIGHLIGHTS OF INITIAL 0012/PAPA TESTS

Some preliminary results from the July 1990 tests will be discussed. The data reduction for the 1991 tests is currently in progress. For these tests the plunge mode frequency was 3.40 Hz with a damping of 0.0017 (fraction of critical damping). The corresponding pitch frequency and damping were 5.18 Hz and 0.0008. The PAPA assembly was balanced such that the pitch axis and the center of gravity were both at midchord.

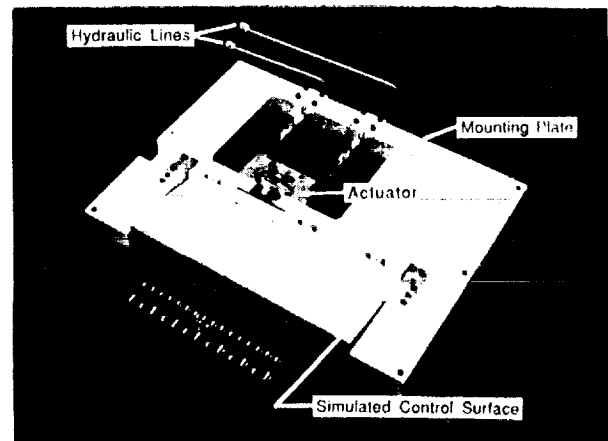


Figure 11. Miniature hydraulic actuator prototype in test fixture.

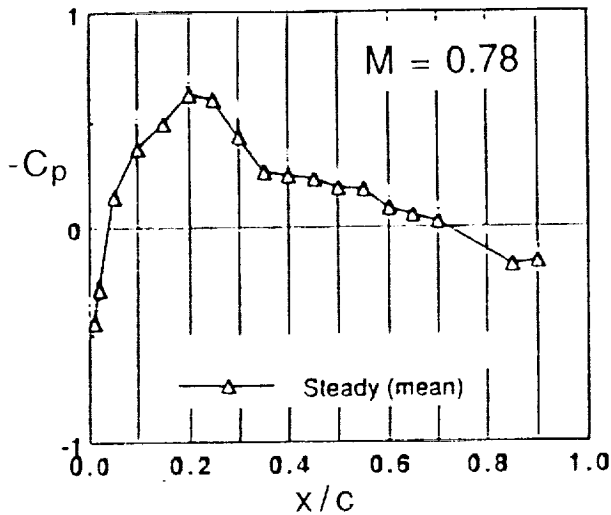


Figure 12. Steady pressure distributions for upper surface at 60 per cent span and zero angle of attack.

**7.1 Steady Pressure Measurements**

The test program included measuring pressures on the model with the PAPA rigidized to prevent pitch or plunge motion. A systematic schedule of Mach numbers and angles of attack up to 4° was run at a value of dynamic pressure near that of flutter, 140 psf (6.70 kPa). This technique should permit evaluation of the static pressure versus the mean pressure during flutter, and the basic unsteadiness of the flow over a stationary model. A sample upper surface pressure distribution is presented in figure 12 for M = 0.78 and for sixty per cent span. For this Mach number a shock is evident near thirty per cent chord. Dynamic data analysis for such conditions should also give an indication of buffet conditions.

**7.2 Flutter at Zero Angle of Attack**

The flutter boundary measured at zero angle of attack is shown in figure 13. The conventional flutter boundary is given by the square symbols. An unusual trend of an increase in flutter dynamic pressure with Mach number is shown which is a result

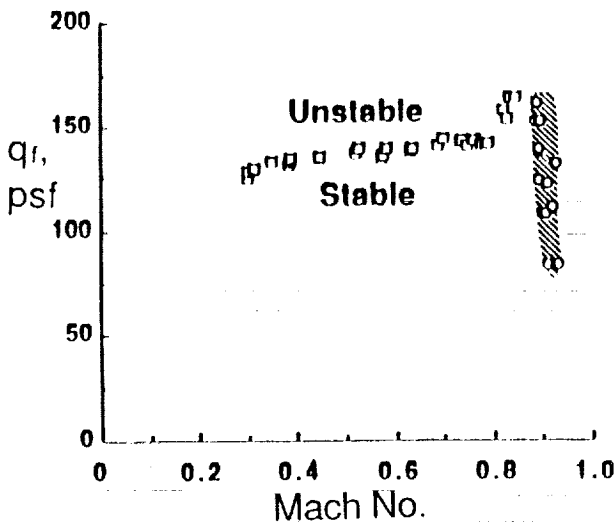


Figure 13. Measured flutter boundaries for 0012/PAPA model at zero angle of attack. (Note 100 psf = 4.79 kPa)

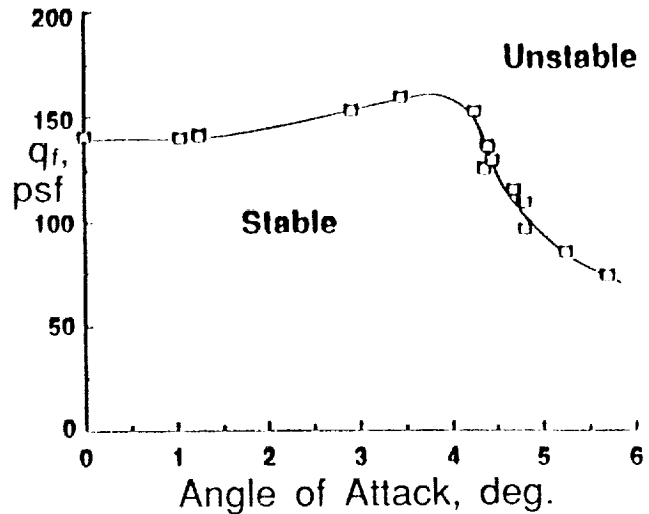


Figure 14. Flutter boundary variation with angle of attack for M = 0.78. (Note 100 psf = 4.79 kPa)

of the aeroelastic parameters of this system. There is a small dip near M = 0.78 and a rapid rise near M = 0.80. Note that the boundary is well defined with a large number of flutter points and relatively small scatter.

In addition to the conventional flutter boundary, a flutter instability involving a nearly pure plunging motion was encountered over a narrow Mach number range from about M = 0.88 to 0.92 as shown by the circular symbols and the cross hatched region (fig. 13). At low dynamic pressures, both the start and end of flutter could be defined, but at the higher dynamic pressures, the motion became so large that only the start of flutter could be determined. Strong shock-induced separation is encountered for this Mach number range. An instability of similar characteristics was also reported for a transport type wing in reference 12.

**7.3 Flutter at Angle of Attack**

The variation of the flutter boundary with angle of attack is shown in figure 14 for M = 0.78. The flutter dynamic pressure shows a small increase with angle of attack for angles up to four degrees. Above four degrees, a rapid decrease in flutter dynamic pressure occurs. Flutter near five degrees has been shown by tufts to involve shock induced separation and reattachment during the cycle of motion. This type of study is difficult to perform on the usual aeroelastic models without exceeding allowable load limitations.

**7.4 Unsteady Pressures Measured During Flutter**

A sample of a measured time history at a flutter point at M = 0.78 and zero angle of attack is given in figure 15. Pitch and plunge motions are shown along with the corresponding unsteady upper surface pressure measurements at x/c = 0.25. The flutter frequency is readily apparent in the pressure, and for this location appears to be nearly in phase with the plunge motion.

The range of unsteady pressure measurements can be visualized by plotting the mean, minimum, and maximum of the pressures as shown in figure 16. For this example, there appears to be only small changes in pressure near the trailing edge of the airfoil, but large changes in the forward portion.



Harmonic or Fourier analysis of the unsteady pressures are performed to determine the amplitude and phase of the first harmonic of the pressure sensed by each transducer. Data of this type are shown in figure 17 for the sixty per cent span section at  $M = 0.39$  and  $M = 0.78$ . The phase is referenced to plunge displacement. The magnitude of the pressures at  $M = 0.39$  display a typical subsonic pressure distribution with a strong peak at the leading edge and decreasing rapidly near the trailing edge. The upper and lower surface pressures are essentially identical. The upper and lower surface phases differ by  $180^\circ$ , as expected, and vary only slightly from leading to trailing edge. At  $M = 0.78$ , (fig. 17b) the magnitude shows a strong forward loading ahead of the shock wave near  $x/c$

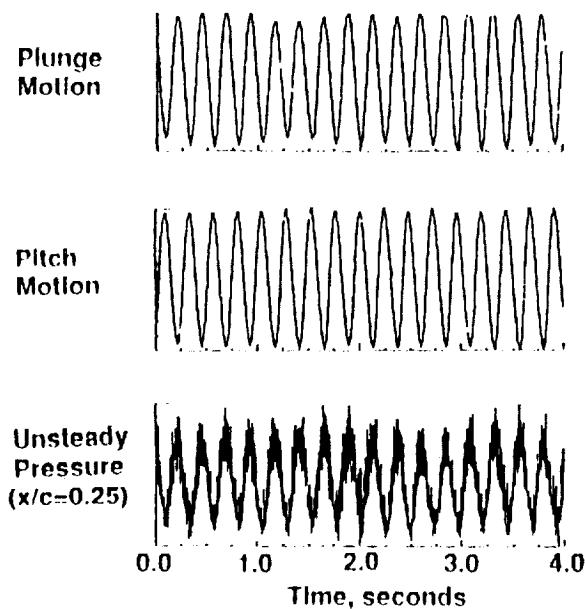


Figure 15. Sample time histories at flutter,  $M = 0.78$ ,  $f = 4.15$  Hz.

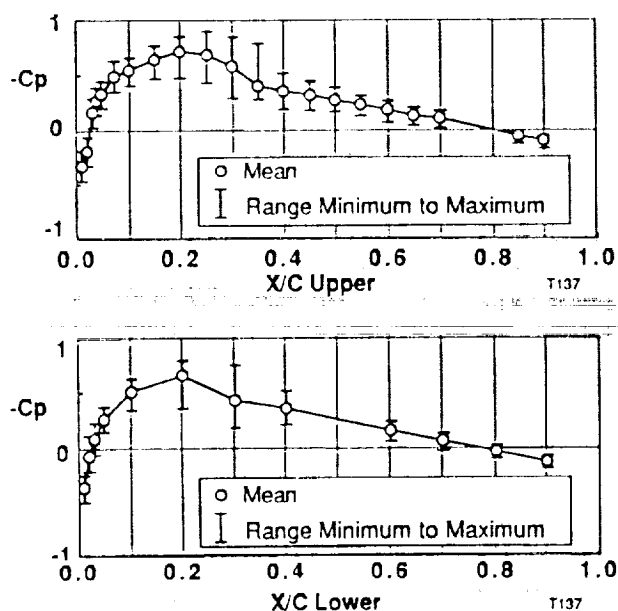
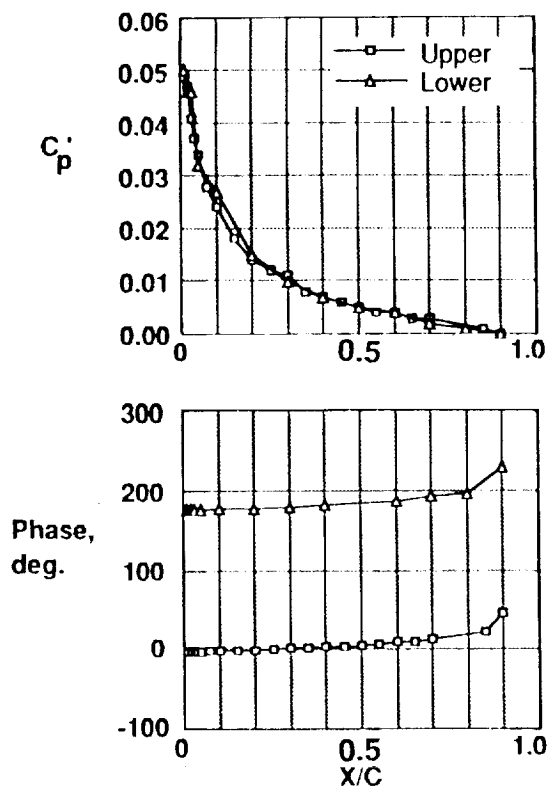
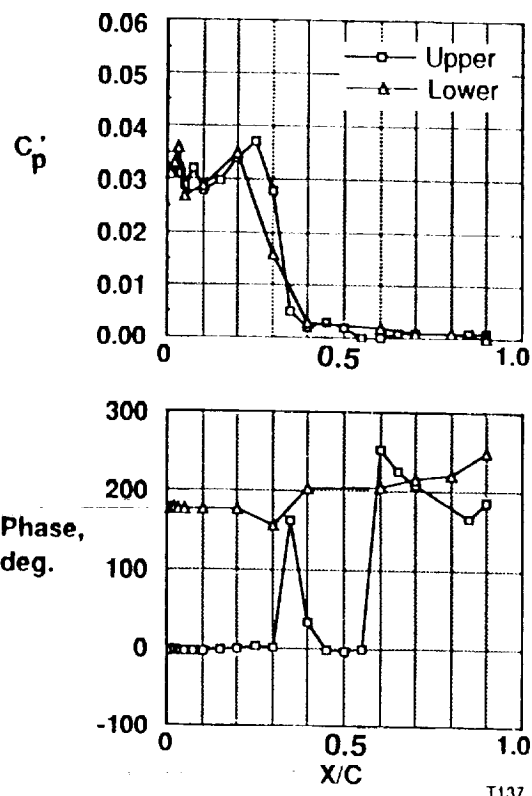


Figure 16. Mean, maximum, and minimum of measured pressures during flutter,  $M = 0.78$  and zero angle of attack.



(a)  $M = 0.39$ ,  $f = 4.50$  Hz.



(b)  $M = 0.78$ ,  $f = 4.15$  Hz.

Figure 17. Magnitude and phase of measured pressures during flutter at zero angle of attack.

= 0.30, and little loading aft of the shock. The phase (fig. 17b) also shows a rapid variation through the shock and some difference in trends near the trailing edge where the magnitude is small. Data of this type can be displayed by the TDT data acquisition system in nearly on-line fashion. Results such as these including the measured flutter modal amplitude and phase information should be valuable in CFD code calibration efforts.

**7.5 Flow Visualization**

Tufts and shear sensitive liquid crystals have been used to give some indication of surface flow features. White tufts have been used on a model painted flat black to indicate separated flow features. These features are recorded with a video camera for later analysis. The liquid crystals, which are normally used for transition detection, have been found to indicate surface features such as shock waves much like oil flow techniques. These techniques have been applied to the NACA 0012 wing on the PAPA and will serve as a qualitative guide in the CFD code calibration efforts.

**8. OTHER BENCHMARK MODELS**

**8.1 HSCT Aeroelastic Model**

As indicated in figure 2, the Benchmark Models program includes a high speed civil transport (HSCT) model scheduled to be tested in January 1994. This model is in the preliminary or conceptual design stage at this time. It is planned as a flexible model in contrast to the rigid PAPA models previously described. Currently, the design is a half model, wall mounted, and has a control for excitation of the aeroelastic modes prior to flutter. An extensive number of unsteady pressure transducers and accelerometers will be used. The total number of channels will be near the 256 channel limit of the facility.

**8.2 Thick Circular-Arc Airfoil Model**

The Benchmark Models Program involves both highly instrumented models for CFD calibration work and simple models for concept exploration or a brief look at interesting physical

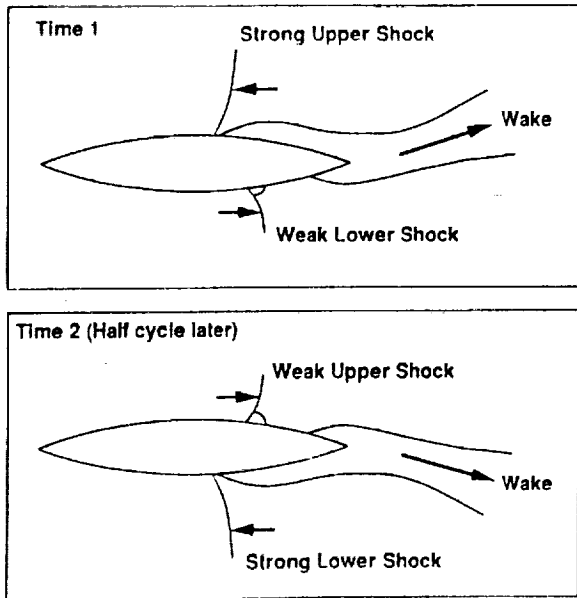


Figure 18. Sketch of transonic shock-boundary layer oscillation on circular-arc airfoil.

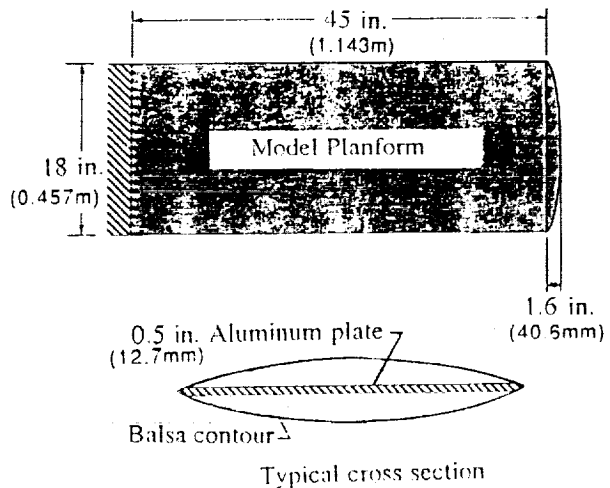


Figure 19. Sketch of wing with thick circular-arc airfoil.

phenomena. One simple model that was built and tested was a flexible rectangular wing with an 18 percent circular-arc airfoil section.<sup>3</sup> The model was built to study the dynamic response of a flexible wing to transonic shock-boundary layer oscillations that occur on thick circular arc airfoils over a small range of Mach numbers. The conditions for this oscillation are illustrated in the sketch of figure 18. As Mach number is increased subsonically, the strength of the shocks terminating the supersonic region on the fore part of the airfoil increases. Initially, a small separation zone occurs at the foot of the shock and at the trailing edge. As the Mach number is further increased, the flow over the airfoil becomes fully separated behind the shockwave. On the thick circular-arc airfoils, near the Mach numbers where the transition from partial to fully separated flows takes place, there is a Mach number range of about 0.04 where the flow alternates antisymmetrically from partially attached to fully separated flow. This occurs with large pressure changes yielding an alternating lift coefficient of about 0.10 at a high frequency ( $k = \omega c/2V$ ) of about 0.50.

The model planform and cross section are sketched in figure 19. The central portion was a 0.50 inch (12.7 mm) aluminum flat plate with bevelled edges. Balsa wood was glued to the plate

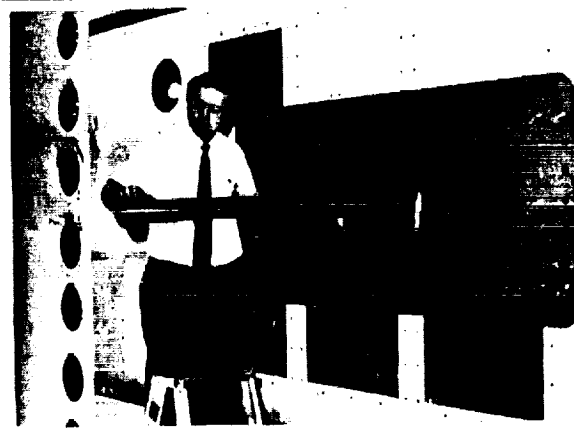


Figure 20. Wing with 18% circular-arc airfoil mounted in the TDT.

with the grain running spanwise and formed to an 18 percent circular-arc section with sharp leading and trailing edges. The root of the plate of the model was clamped in a near cantilever fashion to a turntable in the wall of the tunnel. A small splitter plate of about 6 feet (1.83 m) in length and 3 feet (0.914 m) high was used to keep the root of the model outside the tunnel wall boundary layer. The model is shown mounted in the TDT in figure 20. Transition was fixed at ten percent chord.

For the configuration presented herein, the first bending frequency was 7.8 Hz, and a 3rd bending mode that involved splitter plate motion was at 92 Hz. The splitter plate was attached to the wing mounting bracket and coupled with the wing in this case for the higher frequency modes.

## 9. HIGHLIGHTS OF THE TESTS OF THE CIRCULAR ARC WING

### 9.1 Character of the Measured Results

The overall character of the results is illustrated in the short segment of time histories presented in figure 21. For low Mach numbers, the first bending mode responded at its frequency (7.8 Hz) with random beating or bursts of motion typical of a buffeting response (fig. 21,  $M = 0.751$ ). As Mach number was increased, the buffeting of the first bending mode increased and nearly constant amplitude response in the third bending mode at approximately 90 Hz was also observed (fig. 21,  $M = 0.781$ ). Further small increases in Mach number resulted in little change (fig. 21,  $M = 0.795$ ), until slightly above a Mach number of 0.80 no further response of the third bending mode was apparent (fig. 21,  $M = 0.819$ ). Bending response was obtained only in the 1st and 3rd bending modes and not in the 2nd bending mode.

The root-mean-square (RMS) responses were calculated after low-pass and high pass filtering and are shown in figure 22 in nondimensional form. The responses increase rapidly near  $M = 0.76$  and decrease rapidly again near  $M = 0.80$ . This corresponds closely to the Mach number range of the shock-boundary layer oscillations for the 18% circular arc airfoil.<sup>13</sup>

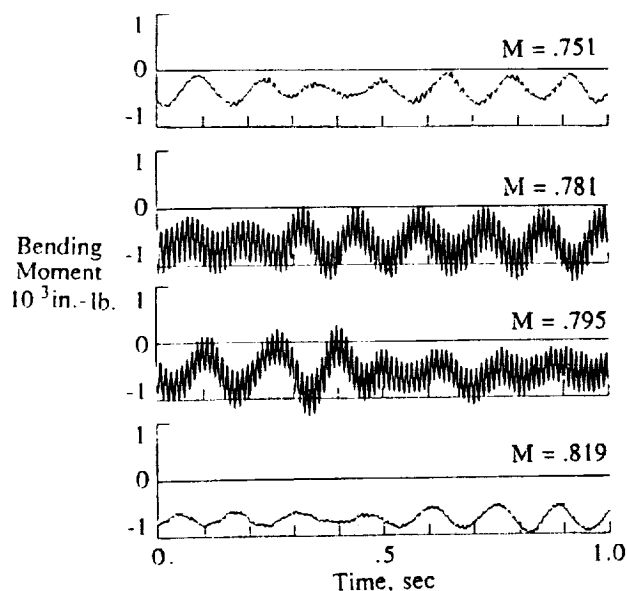
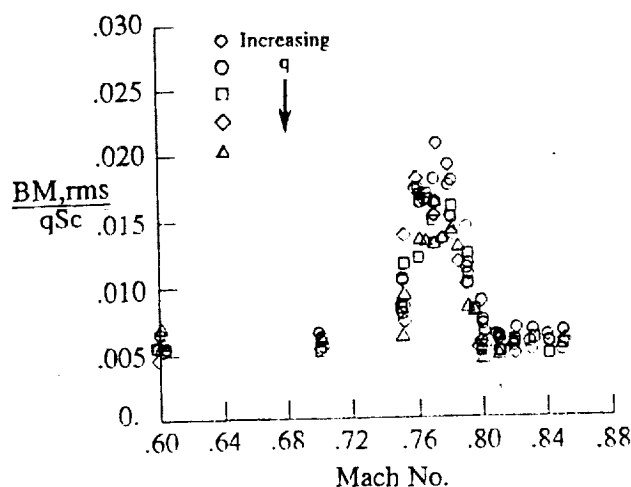
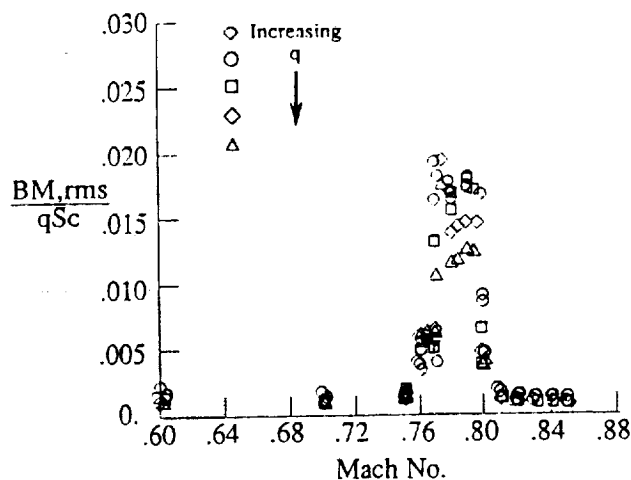


Figure 21. Sample of time histories of bending moment



(a) Filtered low frequency response (1st bending).



(b) Filtered high frequency response (3rd bending).

Figure 22. Bending moment response measurements for several wind tunnel pressures.

Similar levels of RMS response are obtained for both modes. These results indicate that the region of shock-boundary layer oscillations leads to a buffeting condition on this wing for the 1st bending mode which was well removed in frequency for the aerodynamic oscillations, and also leads to a limit-cycle oscillation for the 3rd bending-like mode. The dimensional frequency for the shock boundary layer oscillation is calculated to be 93 Hz, based on  $k = 0.5$ , which is quite near the 3rd bending frequency. Large effects of the transition strip and removal of the splitter plate were also found.<sup>3</sup>

#### 9.1.1 Liquid crystal pattern

During this test, shear-sensitive liquid crystals were used to visualize surface flow phenomena in the spirit of oil flows. A liquid crystal pattern for  $M = 0.82$  is shown in figure 23. At this Mach number the flow behind the shock should be non-oscillatory and fully separated. The light line gives an indication of the shock location and shows a nearly constant chord location over much of the span. However a strong tip effect with a complex flow pattern is visible.

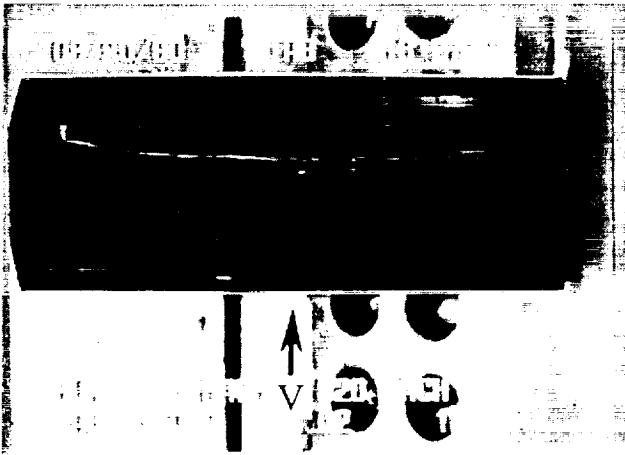


Figure 23. Liquid crystal pattern on wing with circular arc airfoil,  $M = 0.82$ .

9.1.2 Effect of spanwise strip

A spanwise wire located aft of the shockwave was shown to be a good fix or suppressor of the shock-boundary layer oscillations.<sup>14</sup> In the present study, a 0.25 inch (6.4 mm) square strip with rounded corners was taped to the surface at  $x/c = 0.75$  on both upper and lower surfaces.<sup>3</sup> The low and high frequency results are shown in figure 24. The high frequency oscillations are effectively suppressed. However the trend for the low frequency buffeting shown (fig. 24) persisted at lower Mach numbers and a large increase in buffeting levels was obtained. A data point (not shown) at  $M = 0.43$  gave a bending moment coefficient of 0.033 which is a pronounced increase in buffeting level. In summary, the spanwise strip eliminates the high frequency oscillation, but has the strong and undesirable side effect of increased subsonic buffeting response.

9.1.3 Effect of vortex generators

The Wheeler wishbone-type vortex generators were applied to the circular-arc model in an effort to suppress the aerodynamic oscillations as shown in figure 25. These vortex generators are normally used as sub-boundary layer devices, but here they were 0.100 inch (2.5 mm) and 0.96 inch (2.4 mm) high and

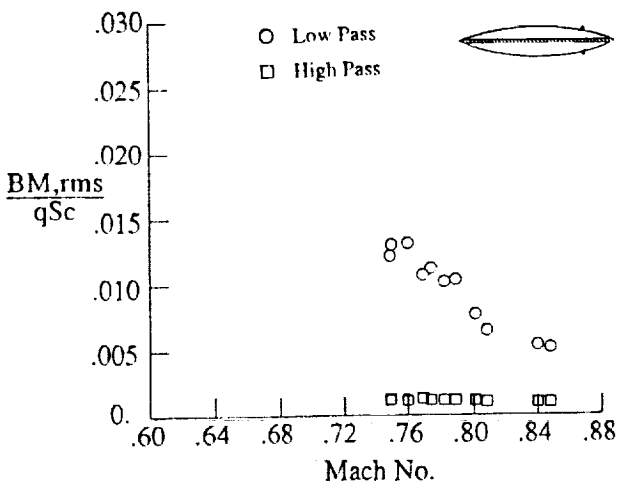


Figure 24. Filtered measured bending moment with spanwise strip.

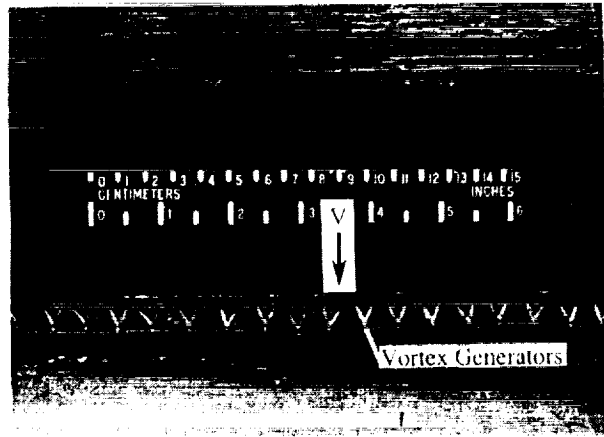


Figure 25. Photograph of wing with vortex generators, configuration 1.

were higher than would be considered sub-boundary layer devices. These were applied at 60% chord. The low and high frequency test results are shown in figure 26. The high frequency oscillations were effectively suppressed, but the low frequency buffeting grew in the transonic range. A large flutter-like response, with a frequency near the 1st bending frequency was encountered near  $M = 0.80$  (fig. 26). Moving the vortex generators forward to 45% chord resulted in some reduction of the low frequency buffeting, but did not satisfactorily suppress the high frequency mode.<sup>3</sup> This type of vortex generator appears to have potential for alleviating the dynamic effects of shock boundary layer interaction, but must be carefully designed and further development is required.

Experience with these efforts to eliminate the shock boundary layer oscillations indicates that fixes derived on rigid models need to be tested on a dynamic model to verify that unsatisfactory side effects are not induced.

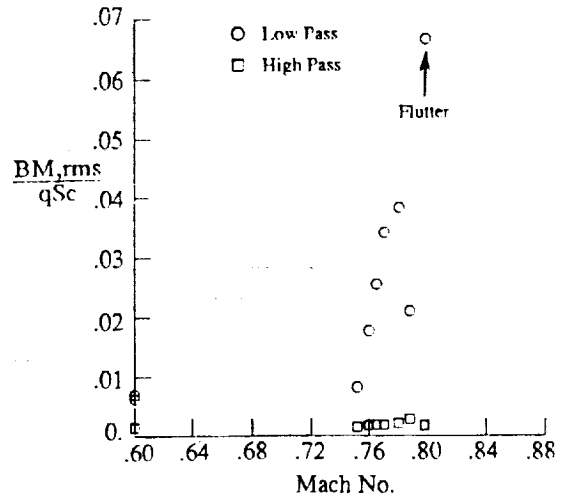


Figure 26. Filtered measured bending moment with vortex generators, configuration 1.

## 10. CONCLUDING REMARKS

The NASA Langley Research Center Structural Dynamics Division Benchmark Models Program has been described. This program consists of about two tests per year over a five year period. The primary purpose is to obtain data for calibration or validation of modern CFD codes for aeroelastic analysis. In addition, the goals of increased understanding of the physics of unsteady flows, and the developing of a data base for empirical design are also included. The overall plan has been described and some of the highlights of the initial test presented including initial tests of flutter of a rigid wing on the flexible PAPA system, and tests of a simple wing with a thick circular arc airfoil have been carried out. Further tests are proceeding and it is hoped that in the very near future additional data suitable for CFD validation efforts will be available.

## 11. ACKNOWLEDGEMENTS

We wish to acknowledge the significant assistance of Clifford J. Obara of the Flight Research Branch of NASA Langley with the application of liquid crystals, and of John C. Lin of the Low Turbulence Pressure Tunnel Section of NASA Langley with vortex generator application.

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REPORT DOCUMENTATION PAGE			Form Approved OMB No. 0704-0188	
Public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Washington Headquarters Services, Directorate for Information Operations and Reports, 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302, and to the Office of Management and Budget, Paperwork Reduction Project (0704-0188), Washington, DC 20503.				
1. AGENCY USE ONLY (Leave blank)	2. REPORT DATE December 1991	3. REPORT TYPE AND DATES COVERED Technical Memorandum		
4. TITLE AND SUBTITLE The Benchmark Aeroelastic Models Program - Description and Highlights of Initial Results			5. FUNDING NUMBERS 505-63-50-12	
6. AUTHOR(S) Robert M. Bennett                      Bryan E. Dansberry Clinton V. Eckstrom                      Moses G. Farmer Jose A. Rivera, Jr.                      Michael H. Durham				
7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES) NASA Langley Research Center Hampton, VA 23665-5225			8. PERFORMING ORGANIZATION REPORT NUMBER	
9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES) National Aeronautics and Space Administration Washington, DC 20546-0001			10. SPONSORING/MONITORING AGENCY REPORT NUMBER NASA TM-104180	
11. SUPPLEMENTARY NOTES Presented at the AGARD Structures and Materials Panel Specialists' Meeting on Transonic Unsteady Aerodynamics and Aeroelasticity, in San Diego, California, October 9-11, 1991, Paper No. 25.				
12a. DISTRIBUTION/AVAILABILITY STATEMENT Unclassified - Unlimited Subject Category 02			12b. DISTRIBUTION CODE	
13. ABSTRACT (Maximum 200 words) The Structural Dynamics Division of NASA Langley Research Center has implemented an experimental effort in aeroelasticity called the Benchmark Models Program. The primary purpose of this program is to provide the necessary data to evaluate CFD codes for aeroelastic analysis. It also focuses on increasing the understanding of the physics of unsteady flows and providing data for empirical design. This paper gives an overview of this program and highlights some results obtained in the initial tests. The tests that have been completed include measurement of unsteady pressures during flutter of rigid wing with a NACA 0012 airfoil section and dynamic response measurements of a flexible rectangular wing with a thick circular arc airfoil undergoing shock-boundary layer oscillations.				
14. SUBJECT TERMS Aeroelasticity and Flutter; Wind Tunnel Testing; Unsteady Transonic Aerodynamics			15. NUMBER OF PAGES 12	
			16. PRICE CODE A03	
17. SECURITY CLASSIFICATION OF REPORT Unclassified	18. SECURITY CLASSIFICATION OF THIS PAGE Unclassified	19. SECURITY CLASSIFICATION OF ABSTRACT Unclassified	20. LIMITATION OF ABSTRACT	